

Development of Green Bi-Propellant Stage

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Abstract

Rocket engine propulsion system has a crucial role to play in the environmental safety concerns all over the world. The criticality in handling hazardous propellants and non-environment friendly combustion products are always a challenge for the space community to avoid hazard/pollution products. Green propulsion stages are the future considering the severe environmental effects of toxic propellants in use for the earth storable stages. The major storable propellants in present day use are Nitrogen tetroxide (N_2O_4), unsymmetrical di-methyl hydrazine (UDMH) and mono methyl hydrazine (MMH), all belong to toxic propellant category. The ease of storage and availability still makes the space sector overlook its dangerous side. A study has been carried out to replace an existing toxic earth storable stage with a green bi-propellant stage. Hydrogen peroxide (H_2O_2) based propulsion systems can cater to such category. A preliminary assessment on selection of optimum propulsion parameters, outcome of mission studies, effect of propellant composition and criticalities of a Green Bi-Propellant Stage (GBPS) are presented in this paper

Keywords—specific impulse, payload, mixture ratio, area ratio

I. INTRODUCTION

Worldwide launch vehicles use different propulsion systems/ stages depending on various mission requirements. The key parameters to be considered are selection of propellants, performance, good structural factor and cost benefits. In addition, the current development in propulsion technologies aims at pollution/hazard free propulsion known as “green propulsion” considering the severe impact on the environment due to toxic propellant usage. The currently used major propellant combinations are earth storable propellants (N_2O_4 /UDMH, MON-3/MMH), cryogenic propellants Liquid Oxygen/Liquid hydrogen (LO_2/LH_2) and semi cryogenic propellants (LO_2 /RP1 (Rocket Propellant-1)), LO_2 -CH₄ (Methane)). The advantage of earth storable propellants is that the system is less complex to handle but has the disadvantages of lower specific impulse and causes hazardous environment. The cryogenic propellant provides higher performance, but the system is complex in nature & costly. The semi-cryogenic engine under development has performance lower than cryogenic ones & costlier. In view of the existing stages available/under development within ISRO, it was intended to explore an alternate propulsion system which has higher performance characteristics than earth storable and can fall under green propulsion category with significant cost benefits compared to other stages. One of the suitable candidates is found to be H_2O_2 (hydrogen peroxide)/RP1 propellant combination which is easy to handle like earth storable ones, gives an overall good performance compared to earth storable and comparable with semi-cryogenic engines. Unlike earth storable propellants, this propellant should be made hypergolic or suitable igniter need to be used. Catalytic decomposition of H_2O_2 can also

make it above auto ignition temperature for proper combustion with RP1/Isrosene. The most important part is it falls under green propulsion technology and offers reduction in propellant cost significantly.

II. SELECTION OF INPUTS

Even though this study aims at development of a new H_2O_2 /RP1 stage, the design inputs were selected considering its application in the existing launch vehicle. In-order to evaluate the advantages/performance of the proposed stage configuration, different case studies were carried out considering it as a replacement to existing stages. Based on the studies it is found that replacement of existing second stage of PSLV (Polar Satellite Launch Vehicle) shows a payload gain (based on mass and specific impulse sensitivity of existing stages).

The following considerations were taken in to account for finalization of stage parameters for existing launch vehicle

- Proposed stage shall deliver higher ΔV compared to existing stage.
- Proposed stage burn duration shall not be much lower compared to existing stage.
- Higher specific impulse and lower or equal structural factor compared to corresponding existing stage.
- Initial acceleration & Thrust to weight ratio of stage is not varying to a very large extent from existing stage considering the trajectory and mission design.

Based on different case studies carried out, the following parameters are worked out for the proposed GBPS.

TABLE 1 Design Inputs

Parameter	Value
Thrust, kN	700
Chamber pressure, bar	50
Propellants	98% H_2O_2 /RP1
Operating cycle	Gas generator

III. SELECTION OF AREA RATIO

The nozzle expansion area ratio is selected considering the engine operating regime. For ground lit engines the area ratio (AR) is selected such that no separation occurs. As the area ratio increases the specific impulse (Isp) increases due to increase in thrust coefficient (C_F) but the increase in expansion ratio increases the mass of the nozzle. Hence a tradeoff is usually done for selecting the optimum area ratio. Figure 1 shows the variation of thrust coefficient w.r.t area ratio.

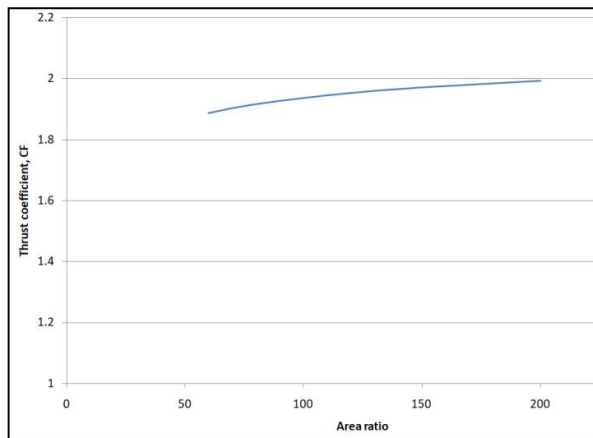


Figure 1 Variation of CF w.r.t AR

Thrust coefficient increases with area ratio. But the corresponding mass increase is significant from AR100 to AR150.

Also, the ΔV provided is estimated for different area ratio & corresponding Isp, keeping all other parameters same. Figure 2 gives variation of ΔV w.r.t area ratio. It can be seen that beyond AR100, there is loss in ΔV . Considering these, the optimum area ratio is 100. But due to the existing envelope constraint of launch vehicle considered, area ratio considered for new stage is 50.

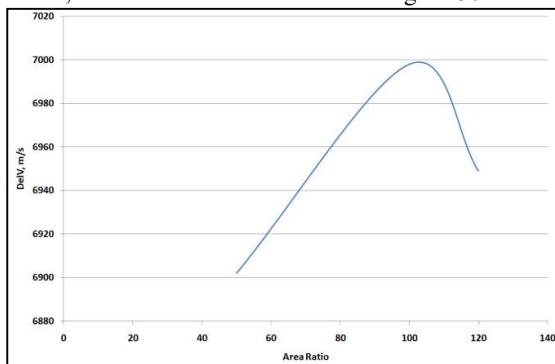


Figure 2 Variation of ΔV w.r.t area ratio

IV. SELECTION OF MIXTURE RATIO

Mixture ratio of an engine decides the propellant requirement which in turn decides the stage mass and plays a key role in the estimation of structural factor of the stage. For every propellant there is mixture ratio corresponds to maximum performance and an optimum mixture ratio considering the mass and thermal aspects. Figure 3 shows the variation of specific impulse and the chamber gas temperature for the propellant combination w.r.t mixture ratio for a given AR.

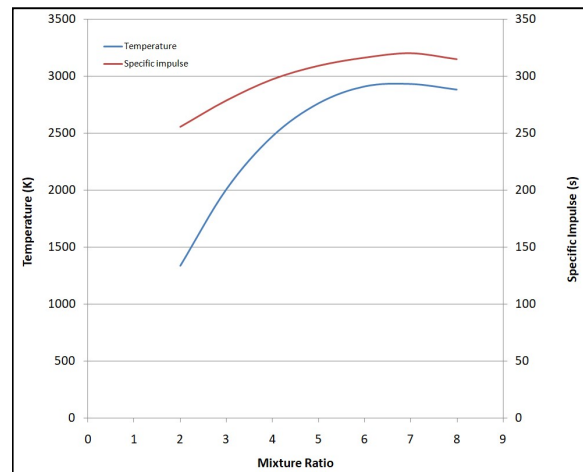


Figure 3 Performance characteristics of H2O2/RP1

The maximum temperature is obtained at MR - 6.8 and maximum specific impulse at 7.3. Considering higher specific impulse and lower chamber temperature, mixture ratio selected for the stage is 7.3.

V. SELECTION OF CHAMBER PRESSURE

Chamber pressure selection depends on the weight, specific impulse and the maximum operating conditions. The lower the P_c the smaller the chamber size. But higher P_c demands higher pump outlet pressure which in turn necessitates higher pressure budget. Hence based on the current engine design for preliminary assessment a chamber pressure of 50bar is selected considering the heritage of existing engines.

VI. STAGE PARAMETERS

The engine and stage performance parameters were estimated for the given input. An efficiency of 0.99 and 0.985 has been taken in to account for characteristic velocity and thrust coefficient respectively. The properties of propellants were taken as reported [9].

Design of subsystems were carried out taking in to account the design criteria to be followed for liquid rocket engines [2][5]. Combustion chamber parameters were worked out considering an L^* of 0.9m. The contraction ratio selected is 2.5 and nozzle area ratio 50. Contour nozzle is considered for the optimum design. Cooling of chamber can be regenerative or ablative cooling. Injector selected is co-axial type. The pressure drop across the injector is maintained as 15-20% of the chamber pressure. The turbo pumps considered are of centrifugal types and is powered by turbine which is driven by the hot gas from the generator output. Another method to drive the turbine is using the decomposed H_2O_2 . Preliminary estimation of component masses was carried out based on empirical relations. The same has been validated with existing measured stage mass. The percentage variation in estimation and measured value is incorporated for the new stage component mass calculation. Table 2 shows the major stage parameters for the GBPS.

TABLE 2 Stage parameters

Parameter	Value
Propellant loading, kg	41202
Fuel tank volume, m ³	6.37
Oxidiser tank volume, m ³	27.25
Diameter of stage, m	2.8
Length of stage, m	~12
Thrust to weight ratio	1.2
Velocity imparted (ΔV), m/s	3331
Specific impulse, s	307
H ₂ O ₂ mass flow rate, kg/s	204.14
RP1 mass flow rate, kg/s	27.96
Chamber diameter, m	0.491
Throat diameter, m	0.311
Overall length of thrust chamber assembly, m	3.2
Fuel injector diameter, m	0.0028
Oxidiser injector diameter, m	0.0066
Momentum ratio	0.2
Oxidiser pump speed, rpm	7613
Fuel pump speed, rpm	23468
Total power, kW	~1.7
Stage mass**, tonnes	~5.7

**considering all the inert masses [10]

Mission studies with GBPS shows a payload gain of ~103kg for the PSLV and lower propellant cost provides a saving of ~Rs. 400L.

VII. EFFECT OF PROPELLANT COMPOSITION

The rocket propellant grade hydrogen peroxide can be developed with different composition. The performance parameter also varies based on the composition selected due to the change in combustion characteristics. Table 3 shows the effect of propellant composition on the engine performance for a given mixture ratio and nozzle geometry. As evident the performance increases at the cost of higher combustion temperature which necessitates critical design considerations for chamber.

TABLE 3 Effect of propellant composition

Propellant	Combustion temperature, K	Theoretical Specific impulse (Vacuum), s
90%H ₂ O ₂ /RP1	2790	304
95%H ₂ O ₂ /RP1	2873	311
98%H ₂ O ₂ /RP1	2922	315

VIII. CRITICALITIES

The major challenge reported in literature is storage of hydrogen peroxide for long time. The stability is defined by Active Oxygen Loss (AOL). It is defined as a value based on a test at a specific temperature and duration for a combination of specific fluid properties and material with any or no surface treatment or processing. Commonly test is done at 66°C for one week. A term called stability is also used to differentiate the material. Materials are classified based on these two parameters as Class 1 to class4. Based on several test conducted the material in the class 1 category

is only suggested for long term storage. In space, experience of hydrogen peroxide storability is available from the 1960's space missions, Syncom II, Syncom III, and the Early Bird satellite. All of them used a monopropellant hydrogen peroxide propulsion system. In particular Early Bird showed in orbit a lifetime of 3 years but was estimated to have a useful life of 5 years. In Early Bird, Aluminum 1060 H-12 was used for tank shells, 1060F for components and lines. Kelf & Teflon coated stainless steel were used for valve parts. In addition, new class 1 Aluminium alloys reported are RX5000, C557, 7X0X. Other spacecraft with hydrogen peroxide propulsion systems were HS303A & ATS. [3]. Stainless steel 316 is also found to be suitable for the H₂O₂ storage. NASA suggested a guideline of material with AOL <0.2% (Active Oxygen Loss) per week can be considered as a suitable candidate for hydrogen peroxide storage. Metals need to be passivated before its use for storage [7]. Literature provides guidelines for better storage such as surface to volume ratio less than 0.5/cm, higher concentration, lower storage temperature (<37°C), purity of H₂O₂(cleaner is better) etc.

IX. CONCLUSION

Study was carried out for developing a new green bi-propellant stage for launch vehicle/spacecraft propulsion system. H₂O₂/RP1 propellant combination provides better specific impulse for the selected mixture ratio and nozzle geometry and falls under green propulsion category with significant cost benefits compared to existing earth storable stages of ISRO (Indian Space Research Organisation). Stage specification is worked out considering it as a replacement of existing stage to evaluate the advantages/performance. One of the options considered is to replace existing second stage of PSLV with new stage. Several case studies have been carried out to finalise the design inputs which gives payload gain with minimum change in existing vehicle configuration & structural factor. Mission study shows a payload gain of 103kg with the new stage. Design of stage and engine parameters were carried out. In addition to the payload gain, the propellant combination gives a cost saving of ~Rs. 4.0Cr compared to existing propellant combination. This stage can also act as a replacement for other existing earth storable stages. Also, if the constraint of envelope is not considered, for a new vehicle, this stage can provide a specific impulse ~320s. Criticalities like compatible material, storage and handling shall be considered during development. Considering the frequency of launches, developing such a stage with higher performance compared to existing earth storable stages, replacing the hazardous propellants with green propellant and a stage that provides significant cost benefits will be advantageous for the organization in its future missions.

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FUTURE WORK

Considering the advantages of the proposed stage in terms of payload gain and reduced environmental hazards, detailed

stage engineering is planned. Ignition methodology through catalytic decomposition of H_2O_2 or making the oxidizer hypergolic by suitable additives are being explored.

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